SATELLITE CONSTELLATION LAUNCH, DEPLOYMENT, REPLACEMENT AND END-OF-LIFE STRATEGIES

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Abstract

The considerable surge in satellite constellations has brought to the fore the imperative need for an efficient satellite constellation management plan. To address this emerging need, GMV has analyzed the possible strategies for constellation launch, set-up, replacement of failed satellites and end-of-life policy. The constellation launch and deployment has been divided into the launch site and launcher selection, the evaluation of the injection and the transfer strategies, and the set-up phase. The main replacement strategies investigated are based on in-orbit spares, spare satellites in parking orbits and spare satellites on the ground. Finally, end-of-life policies for LEO, MEO and GEO satellites are presented. As such, the analysis accomplished encompasses most of the fundamental phases of constellation life cycle.

A representative constellation of small satellites has been taken into account to assess the effectiveness and the commercial viability of the strategies outlined. This study case has been handled using in-house software tools and algorithms.

Key words: Launcher Selection, Orbit Transfer, Satellite Constellation Deployment, Constellation Replacement Strategy, End-of-Life Policy.

Introduction

The last few years have seen a virtual explosion of satellite constellation concepts, which have been envisioned for a broad range of traditional and new applications. Initially starting with navigation and positioning implementations, constellations have branched out into telecommunications for direct telephony, mobile message systems and broadcasting, as well as Earth observation and data collection missions. This trend in space systems has made imperative efficient satellite constellation launch, deployment and system management, fostering new studies and approaches to meet the surge in constellation developments and space transportation.

To address these emerging needs, GMV has analyzed the possible options and trade-offs and has outlined optimal strategies for constellation launch, set-up, replacement of failed satellites and end-of-life policy. The analysis of these key phases of a constellation management plan has been carried out using in-house software tools and algorithms[6]. This software tool kit combines databases, optimization procedures and Monte Carlo simulation techniques.

The first objective of the analysis deals with the review of the different commercial ***launchers*** in terms of launch mass, injection orbit capabilities (single/multiple), multiple spacecraft launch capabilities, fairing capacity, launcher reliability and launch delay. This information is used to select the launcher(s) which can perform the satellite orbit injection. The launcher selection is also strongly affected by the launch site availability and compatibility with the injection orbit constraints (mainly driven by the inclination angle).

The second objective consists of the analysis of optimal ***orbit injection and transfer strategies*** that allow the transfer from the launcher injection orbits to the constellation operational orbits. The possibility of a direct injection into the final orbit has been considered mainly in case of LEO constellations.

The third objective is the analysis and the development of ***constellation set-up strategies***. This objective deals directly with the constellation set-up procedure independent of the launcher. The ideal purpose of the constellation deployment would be to achieve a substantial level of service while arranging the constellation, without having to wait for the completion of the set-up phase. The driving concept is that ideally one would like to achieve some performance level with the very first satellite launched and to increase that level of performance with each spacecraft launched.

To handle the set-up problem, the following primary issues need to be addressed:

- What is the sequence in which the satellites should be placed into the constellation such that an acceptable constellation service is achieved with a minimum number of satellites?
- Single, dedicated or multiple launch/launcher strategies?
- How many s/c per launch should be placed?

Once a constellation launch and set-up phase has been accomplished, the satellites are arranged in the nominal configuration set by the constellation orbital design. The next step within the scope of the constellation mission analysis is related to the evaluation of the ***replacement and spare strategies***. This involves estimating the consequences and the necessary steps to take in case one or more satellites fail to operate.

When one satellite fails to operate, the remaining satellites are required to provide needed services at a comparable or reduced level rather than having a total loss of service. So, in this phase of analysis, it is necessary to answer the following questions:

- What happens to the mission return and to the overall system performance if one or more satellites fail to operate?
- How to perform the satellite replacement?
    - spare satellites in constellation?
    - spare satellites in parking orbits?
    - spare satellites on the ground?
- How to ensure the constellation configuration maintenance?

These are important issues since most of the constellations aim to provide the user with a continuous reliable service or at least with a minimum level of service.

The considerable surge in constellation developments has also brought to the fore the imperative need of an efficient constellation management plan. This surge is mainly fuelled by the explosive demand for commercial telecommunications services. This is, in turn, leading to an escalating GEO satellite deployment and to the introduction of telecommunications constellations in low Earth orbit.

Several LEO constellations, encompassing well over 750 satellites, are currently in development. All these systems will have operational lifetimes of many years (8 to 10 years on average), and they will contribute to a growing debris environment that already presents a significant long-term collision hazard. Most LEO constellations will be deployed in altitude bands of peak debris density, thus generating a collision risk that cannot be neglected.

The above-mentioned considerations have strongly fostered the investigation of possible ***end-of-life policies***. The objective is to identify the best strategy to be implemented at the end of the operational lifetime of a constellation, so as to avoid a disproportional increase in the collision risk for satellites positioned in the same altitude band as the constellation. Ultimately, the worst danger is the possibility that a non-operational constellation triggers a cascading process of collisions in its altitude band. Given these concerns, the identification of effective end-of-life policies emerges as a controversial issue that plays an increasingly important role within a constellation management plan.

This paper presents the possible strategies to handle constellation launch, deployment, replacement of failed satellites and end-of-life policy. The main effort has been devoted to define a general approach to the problem, so as to allow the characterization of a wide range of possible requirements and solutions. The concepts and the requirements emerging from the investigative study have been applied to a representative constellation of small satellites, so as to assess the commercial viability of the strategies outlined. GMV performed the analysis of the launch, set-up, replacement and end-of-life strategies for a Data Collection Constellation for an ESA-ESTEC contract, in collaboration with Alcatel Space Industries (prime contractor).

Launch and Set-up Strategy

Considering the launching and the constellation set-up experience gained by the NAVSTAR/GPS constellation and the recently launched ORBCOMM and Iridium systems, one can clearly identify that the constellation launch and deployment procedures are driving factors for the constellation service. The set-up strategy and the associated launcher selection are directly related to the investment plan and to the revenues. They become significant only when a sufficient number of satellites is in orbit and a suitable market penetration is achieved. In this strategic planning some of the key factors are the launcher selection, the deployment cost and the launch service availability and completeness.

Launch Strategy

In investigating the launch strategy and set-up, one of the first decisions to be made is which launcher(s) to use and how many satellites that launcher can inject into orbit. Of primary importance in determining the launcher(s), is the number of satellites which can be launched with a particular launcher. This number depends primarily on two factors:

- The payload mass which the launcher can inject into the desired orbit
- The volume of the launcher fairing

In order for a launcher to be able to launch a satellite, it must be able to inject the mass of the satellite into its desired orbit (this may be either directly or via a transfer orbit) and the satellite(s) must be able to fit inside the launcher fairing.

The possible launchers which may be used to launch a given satellite are highly dependent on the launch site which is used. Therefore the launch site selection represents a key phase in the launch process. The list of possible launch sites for a specific mission can be obtained based on the desired orbit inclination (which is the driving parameter). The possible launch sites are then passed to the launcher selection phase in order to immediately eliminate launchers which cannot be launched from one of these launch sites. Using this information, the following step entails identifying possible launchers based on the satellite mass, orbit altitude, eccentricity, orbit inclination angle, and optionally the satellite dimensions (length, width and height) and launcher adaptor mass. At this point it is possible to estimate the number of satellites which can be launched with the selected launcher(s), as well as information about the launcher(s).

In order to deal with the launch strategy, three databases have been compiled:

- *Launch Site Database*: it provides information as for the launch site location and the possible inclination angles to inject into from each launch site.
- *Launcher Database*: it contains the company name, launch sites, mass which can be launched to GTO, mass which can be directly injected into GEO (if possible), launch cost in millions of $US, dedicated time-to-launch, and the launcher reliability. There are a total of 28 launchers in the database. They are:
    - Ariane 40, Ariane 42L, Ariane 42P, Ariane 44L, Ariane 44LP, and Ariane 44P
    - Ariane 5
    - Athena 1 and Athena 2
    - Atlas IIAS
    - Delta II and Delta III
    - Long March CZ-2C, Long March CZ-2E, Long March CZ-3, Long March CZ-3A, and Long March CZ-4
    - Pegasus XL
    - Soyuz
    - Proton D-1 and Proton D-1-e
    - Rockot
    - Taurus
    - Titan II, Titan III and Titan IV
    - Zenit 2 and Zenit 3.

  In addition, launcher performance profiles have been determined for injections into LEO orbits. These profiles describe the mass injection capability as function of the altitude and inclination of the injection orbit.
- *Fairing Database*: it details the model type, the dimensions and the volume of each fairing for each launcher.

The following flow chart schematically shows which are the main steps to be performed so as to select a launcher to launch a particular satellite or constellation.

1. Define Mission Requirements
2. Define Constellation and Payload Requirements
3. Political Considerations: where to launch the satellite from, using what launch vehicle
4. Launch Site Selection: identify possible launch sites based on the inclination of the target orbit
5. Launcher-Spacecraft Fit
6. Mass and fairing dimensions
7. Constraint Analysis and Optimization with respect to: launch site, cost, reliability, availability, launcher-spacecraft interfaces, launch opportunities
8. Determination of Candidate Launchers
9. Transfer Strategy necessary from the launcher injection orbit to the operative orbits? If so, evaluate the required fuel budget, add it to the mass of the spacecraft, and start the analysis all over again.
10. Launcher Selection: possible launchers and number of satellites that each can inject into the target orbit.

It is to be noted that trying to cope with all the political factors involved in the selection of launch site and launcher was beyond the scope of the analysis. It was therefore decided to present all of the possible launch sites and launchers and allow the mission analyst to eliminate politically undesirable options at the end.

The launcher survey performed also pointed out that, among the launchers reviewed, only Ariane 5 and

Rockot are capable of launching into different planes. But only a very small angular difference can be achieved between two injection planes in terms of right ascension of the ascending node and inclination. Thus, GMV has assumed that no substantial multiple plane insertion capability is available at the moment and that each launch is targeted to inject all the spacecraft lofted atop the launcher into the same orbital plane. This assumption reflects the state-of-the-art technology in the launch vehicle industry and can be dropped if multiple plane insertion capability becomes feasible.

Study Case

In order to test the effectiveness of the launch and set-up strategy outlined, a Data Collection Mission[10] has been taken into account as representative study case as for constellations of small satellites. The overall mission objective is to gather a wide range of information, elaborate it as required and deliver either the processed or the raw data to the users in a non-synchronous data relay mode.

The primary target area is located at low to mid latitudes in the northern hemisphere (26°-57°), since it is expected that the largest demand, in terms of volume of data to be managed, stems from the highly populated and industrializes areas of the world. In order to meet the requirements expressed in terms of mean revisit time in the areas of interest, the design solution is a Walker constellation of 24 satellites symmetrically arranged in 6 orbital planes that are evenly distributed around the equator. The constellation inclination is 57° and the altitude is 850 km. This configuration allows achieving a mean revisit time lower than 3 minutes with a minimum elevation angle larger than 5°.

The spacecraft used to carry out the mission has a dry mass of 130 kg and a cubic structural configuration, with equal dimensions of 0.8 m. In order to arrange the spacecraft inside the launcher, a proper structural interface is required: the adaptor. As reasonable value, we have assumed that the adaptor mass is 15% of the spacecraft dry mass.

Table 1 presents the most interesting launch vehicles resulting from the selection phase, complete with the relevant information about them. Based on the fundamental assumption that no multiple plane insertion capability is feasible, the most interesting possibilities for a direct injection are the launchers that can inject up to four satellites per launch.

No launcher seems to be suitable to launch exactly four satellites at a time, so as to fill a constellation plane with a launch. However, Taurus can launch five spacecraft and can be used to inject four if necessary.

Among the possible launchers obtained, Athena 1, Pegasus XL and Taurus can be classified as relatively small launchers, which are suitable to inject small payloads into LEO. One observation inferred from the table presented is that the cost per spacecraft launched is rather high if these small launchers are used to inject the satellites. Currently, the small launcher suppliers are supporting a development plan aimed at cost reduction in order to make their vehicles more attractive and competitive, particularly for deploying the emerging small satellite systems.

Table 1: *Selected Launchers for the Data Collection Mission (Launcher data up to May 1999).*

Launcher	Launch Site	Fairing Volume (m^3)	Mass (kg)	S/c per launcher	Launcher Cost ($M)	Cost per s/c ($M)	Launcher Reliability
Athena 1	Cape Canaveral, Vandenberg	10.53	391.	2	16	8	50%: 1/2lnchs
Athena 2	Cape Canaveral, Vandenberg	29.00	1322.	8	21	2	80%: 4/5lnchs
Long March CZ-2C	Jiuquan	26.75	499.	3	15-20	5-6	85%
Pegasus XL	L-1011 Airplane	2.00	250.	1	6-15	6-15	86%: 18/21 lnchs
Rockot	Baikonur, Plesetsk	24.18	1353.	9	13-15	1	100%: 3/3 lnchs
Taurus	Cape Canaveral, Vandenberg	4.74	900.	5	18-20	3-4	New launcher

From the evaluation of the possible launchers for the Data Collection Mission it emerges that the number of choices is narrowed down when the deployment of a constellation of small LEO satellites is taken into account. The most powerful launchers, which have a considerable mass injection capability, do not prove to be advantageous possibilities because they can launch a considerable number of spacecraft into an injection plane (sometimes more than the number of satellites in the whole constellation).

Taking into account only cost considerations, Rockot and Athena 2 emerge as the most advantageous choices (lowest cost per s/c launched). These launch vehicles can inject more than four satellites per launch, so they are considered very good options for an indirect injection, which allows populating various operational orbital planes with a launch.

It is important to underline that the suggested launchers are intended to be used by the mission analyst as guidelines for the evaluation of the launch and deployment strategies. The values of the launcher data displayed are approximations only, and for more precise and detailed information the launcher manufacturer should be consulted.

Orbit Injection and Transfer Strategies

A *direct injection* of the satellites into the final orbits of the constellation is conducive to carrying out a quick deployment strategy. This allows starting the system operations and revenue stream early in the management plan. The combination of final orbit altitude and inclination, along with launcher capability, determines the feasibility of this injection strategy.

The launch vehicle may require an upper stage to achieve the final orbit, adding to the launch cost. The alternative is to provide sufficient propulsive capability on the spacecraft to perform the propulsive maneuvers needed to reach the final orbit. The impact on the spacecraft in additional propellant and tanks must be carefully weighed in terms of cost and complexity of the spacecraft design and traded against the potentially higher launch cost associated with an upper stage.

Another option is an *indirect injection*: the altitude of the injection orbit is lower than the operational one and the differential effect of the Earth oblateness on the node (due to the altitude difference) can be used to change the node of the satellites and populate several operational orbital planes.

An indirect injection strategy proves to be useful particularly if the launcher selected to perform the orbit injection is capable of launching a number of satellites greater than the number of spacecraft in each of the constellation orbital planes. In this case, the satellites can be launched into a drifting orbit and transferred to their target orbit when the drifting plane and the target plane overlap. The transfer maneuver to be accomplished depends on the propulsion system used. In case of high thrust propulsion systems, the orbit transfer can be considered impulsive and a transfer maneuver can be performed each time the drifting plane crosses one of the target planes of the constellation. In case of low thrust propulsion systems, the transfer maneuver should start before the drift plane crosses the target plane. Since the maneuver cannot be considered impulsive, it is necessary to account for the rotation of the two planes during the raising phase.

Independent of the strategy applied, the orbit transfer must be optimized by taking into account such criteria as the ΔV required (i.e. the necessary fuel budget) and/or the transfer time, and trying to minimize them.

Orbit Transfer Methods

Two methods have been analyzed to perform the orbit transfer: the Impulsive Orbit Transfer and the Orbit Transfer based on J2-Angular-Drift. The basic difference between these two methods consists of the propellant and time resource allocation.

- In the *impulsive orbit transfer* the orbit transfer is carried out by using a sequence of impulsive maneuvers, each characterized by a value of the corresponding ΔV required. In-plane as well as out-of-plane maneuvers are implemented.
- The *orbit transfer based on J2-angular-drift* takes advantage of the secular drift of the right ascension of ascending node (RAAN) due to the J2-term of the terrestrial gravitational potential. This effect is used to create a relative precession motion between two orbital planes so as to obtain the desired RAAN separation.

The basic assumption for the orbit transfer based on J2-angular-drift is that the initial (drift) orbit and the final (target) orbit have the same plane inclination. Because of this and since the initial difference of RAAN is cancelled thanks to the differential drift, only in-plane impulsive maneuvers are necessary to perform the orbit transfer. An optimal low-cost Hohmann transfer is applied once the drift time has elapsed and the two orbital planes have the same RAAN. One-impulse transfer and two-impulse transfer strategies are considered to perform the in-plane maneuvers.

In case of chemical propulsion, the in-plane transfer phase between the drift orbit and the final orbit is small compared to the drift duration (the time on the drift orbit). Therefore, the overall transfer duration is assumed equal to the drift duration. On the other hand, if an electric propulsion system is used, the transfer phase between the drift orbit and the final orbit cannot be neglected, since it represents a significant part of the total transfer duration.

In the most general case, the semi-major axis and the eccentricity of the drift orbit are the keplerian elements that affect its node precession and are the parameters to be determined. A constraint on the maximum transfer time can be imposed and the corresponding ΔV can be computed.

This transfer strategy is conducive to saving the fuel allocated to the orbit transfers, since it allows avoiding out-of-plane maneuvers that are generally very expensive. On the other hand, the duration of the overall transfer phase turns out to be longer than in the case of a transfer strategy based on totally propulsive maneuvers. Thus, the propellant consumption must be traded against the overall transfer time.

Study Case

In principle, it is reasonable to assume that a direct injection into the final orbits of the Data Collection Constellation is feasible. In any case, to evaluate the impact of a propulsive maneuver in terms of propellant mass required (and hence additional mass of the spacecraft to be lofted atop the launcher), GMV has calculated the cost of the impulsive transfer from the injection orbit of the launcher to the final orbits of the constellation.

The altitude of the injection orbit is varied and the propellant mass required for the optimal two-impulse transfer maneuver is computed. The launcher injection orbit is assumed to be circular, with the same inclination angle as the constellation. Thus, the transfer maneuver is devoted only to increase the orbit altitude. The propulsion system is chemical, with a specific impulse of 280 s. The results are presented in a graphical format in Figure 1.

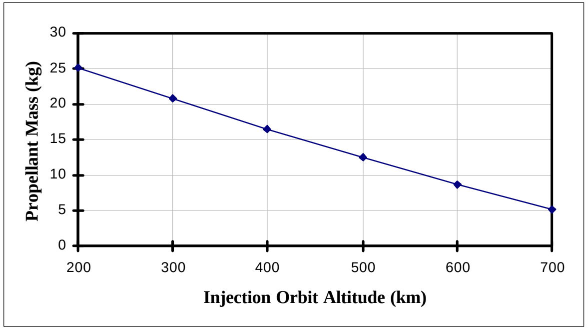

Figure 1: Propellant Mass for the Impulsive Transfer to the Data Collection Orbit

In order to evaluate if any advantage is achieved thanks to a propulsive transfer from a low launcher injection orbit to the orbit altitude of the constellation, GMV has assumed a circular injection orbit at 200 km (a very common LEO injection) and has calculated the necessary propellant to raise the altitude. This additional propellant mass has been added to the spacecraft and the launcher selection process has been investigated again, this time using an increased value for the satellite mass to be launched. The results are presented in Table 2.

Table 2: Comparison of Injection Strategies for the Data Collection Mission.

	Direct injection into 850 km		**Injection into 200 km (followed by an altitude-raising phase)**	
1 s/c launched	*Pegasus XL* 6-15 $M per s/c launched		*Pegasus XL** 6-15 $M per s/c launched	
2 s/c launched	*Athena 1* 8 $M per s/c launched		*Pegasus XL* 3-7 $M per s/c launched	
3 s/c launched	*Long March CZ-2C* 5-6 $M per s/c launched		*Athena 1* 5 $M per s/c launched	
4 s/c launched	*Taurus ** * 4-5 $M per s/c launched		*Taurus ** * 4-5 $M per s/c launched	
8 s/c launched	*Athena 2* 2 $M per s/c launched	*Rockot **** * 1 $M per s/c launched	*Athena 2 **** * 2 $M per s/c launched	*Rockot **** * 1 $M per s/c launched

*Can launch 2 s/c, but is used to launch 1 s/c. ** Can launch 5 s/c, but is used to launch 4 s/c.

*** Can launch 9 s/c, but is used to launch 8 s/c.

The potential advantage of this injection strategy with respect to a direct injection is that some of the launchers selected (particularly Athena 1 and Pegasus XL) increase their mass injection capability. This entails a potential reduction in the launch cost. In any case, the significance of the potential cost reduction is not so high as to lead to the rejection of a direct injection strategy. Undoubtedly, a direct injection is still a very good option.

The feasibility and the cost of an indirect injection strategy can be estimated using the angular drift due to J_2 to assist the transfer to the orbital planes of the constellation.

Figure 2 presents the node precession due to the Earth's oblateness versus the circular orbit altitude. The inclinations selected to parameterize the curves are close to the value of the study case under consideration. The determination of the drift orbit entails the evaluation of its altitude so as to fulfill the mission requirements, particularly in terms of the overall transfer time.

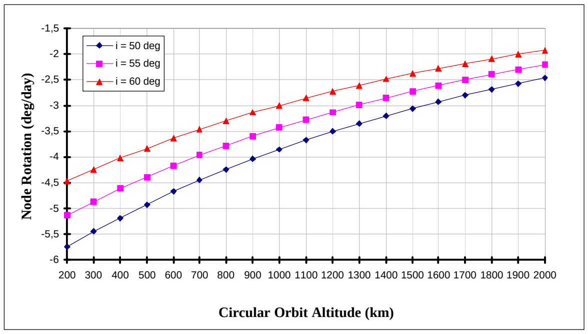

Figure 2: Node precession due to the Earth's oblateness versus the circular orbit altitude

The worst case as for the indirect injection occurs when the initial angular difference between the injection plane and the target plane is equal to the spacing between two constellation planes. Table 3 provides the drift orbit that meets the constraint in terms of the specified drift duration.

Table 3: Indirect Injection of the Data Collection S/c.

Drift Time (months)	Circular Drift Orbit Altitude (km)	Propellant Mass (kg)
2	349.5	18.6
3	499.0	12.5
4	579.6	9.5
5	630.1	7.6
6	664.7	6.3

The indirect injection technique can be an interesting solution if the constellation launch and deployment is carried out using launch vehicles that can inject more than four spacecraft into the desired orbit (e.g. Athena 2, Rockot). These vehicles allow reducing the cost per satellite launched with respect to the launchers selected for a direct injection (Athena 1, Long March CZ-2C, Pegasus XL or Taurus).

Constellation Set-up Strategy

Intermediate Deployments

The need to start the revenue stream flowing as early as possible is a driving force acting upon both the system design and the development schedule. This feature is evident particularly in today's telecommunications satellite systems.

Earth observation constellations generally start data acquisition as soon as sufficient satellites are deployed and operational. The data quality and delivery time provided by a partially deployed Earth observation constellation are coarser than the nominal ones, which can be achieved only when the whole system achieves the full operational status.

In case of navigation and positioning constellations, the deployment strategy varies widely and the constellation typically does not function until all the spacecraft are operational. So, intermediate system arrangements are not generally dealt with.

A considerable interest in partial deployment options arises if we consider the telecommunications constellations that are in a development status. The GEO community is accustomed to launching a single satellite and starting the revenue stream shortly after the system checkout phase has ended. The revenue stream from the market area served by this first satellite can then be used to self-fund any additional GEO satellites called for by the constellation set-up plan. The LEO (or MEO) system dynamics do not allow the same revenue results after the launch of their first satellite, thus forcing the constellation designers to investigate partial deployment options that can achieve a significant service.

The evaluation of the possible partial deployment options proves to be very useful particularly in case of symmetric, inclined constellations (also referred to as "Walker Constellations"), and it allows identifying progressive deployment steps to be traced in the system set-up phase.

A major difference between the inclined and the polar constellations emerges when we investigate the feasibility of intermediate deployments. Polar orbit, single coverage systems must have all satellites or almost all satellites arranged in orbit before continuous service can be provided to customers in moderate latitude bands. This is due to the fact that polar systems are capable of providing the best coverage performance at high latitudes, in concentric rings around the poles. The poles turn out to be the only geographic region where continuous coverage may be achieved soon. Unfortunately, this region is not usually the primary service area in the system's business plan. Therefore, the choices for early revenues available to systems employing polar orbits are limited. The solution that is currently being put forward consists of a rapid deployment campaign. This solution attempts to place the constellation into space via a multi-satellite per launch vehicle strategy carried out within a very short time schedule.

Inclined orbit constellations have at least one more deployment choice available to them. They may be capable of providing continuous service to bands of latitude near the maximum and minimum latitudes of the satellite ground track by deploying only a portion of their full system. This partially deployed constellation could then begin early revenue operations while the remaining planes of the full constellations are populated.

In the attempt to outline a methodology to identify possible intermediate constellations, some criteria have been defined to select the most significant partial deployment options. These criteria are intended to define the compatibility of a partial deployment solution with the configuration of the constellation that must be set up (the full constellation). The following criteria of selection are applied to identify an intermediate constellation:

– the number of orbital planes of the intermediate constellation must be less than the number of orbital planes of the full constellation

– the number of s/c per plane must be less than the number of s/c per plane of the full constellation.

These criteria have been chosen because they are conducive to avoiding plane changes to evolve from the intermediate solution(s) to the final configuration of the constellation. Moving a satellite from an orbit plane to another is an expensive maneuver in terms of impulse and hence of propellant. So it is quite desirable to avoid a system evolution strategy that entails out-of-plane maneuvers. Because of this, plane changes are not envisaged and the above criteria are used to identify intermediate deployment options.

It is to be noted that this procedure allows identifying the main steps of progressive deployment. We cannot rule out that other intermediate steps may prove to be interesting and need to be analyzed in more depth.

Constellation Set-up

The ideal purpose of the constellation deployment would be to achieve a substantial level of service while arranging the constellation, without having to wait for the completion of the set-up phase. The driving concept is that ideally one would like to achieve some performance level with the very first satellite launched and to increase that level of performance with each spacecraft launched. Some analyses have shown that the constellation performance tends to come in plateaus as one inserts one more satellite into each orbit plane of the final constellation. For this reason, constellations with a small number of orbit planes have an advantage in terms of performance build-up over many-planed ones. On the other hand, a smaller number of orbital planes leads to lower degradation. This means, for example, that in a constellation of ten satellites arranged in two planes, if one satellite is lost, by re-phasing the constellation (with little propellant consumption), the performance can be maintained with the level corresponding to an eight-satellite plateau.

To handle the set-up problem, the primary issues to be addressed are the following:

- What is the sequence in which the satellites should be placed into the constellation such that an acceptable constellation service is achieved with a minimum number of satellites?

- Single, dedicated or multiple launch/launcher strategies?

- How many s/c per launch should be placed?

Although the set-up problem can be formulated and handled independent of the launcher(s) selected to perform the satellite orbit injection, obviously there is a correlation between the launcher selection phase and the deployment strategy. The fundamental correlation parameter is the number of spacecraft launched per launch, which depends on the launcher(s) used. Once a launcher has been selected and its mass and injection orbit capabilities have been assessed, we know the number of satellites that it can load and launch. This number is the input needed to develop the deployment procedure, which allows bypassing the explicit definition of the launcher used. In addition, the launcher also drives other important factors, such as the launch cost, the time delay to launch and the launch system reliability.

It is interesting to point out that, since the deployment problem and the launcher selection have been uncoupled, it is possible to change the “logic sequence” of application of the two phases. The mission analyst could cope with the set-up strategy and determine one or more optimal deployment sequences, with the corresponding number of spacecraft launched at each set-up step. Then, the analyst should evaluate if the state-of-the-art launch industry can provide the lift-off capability to carry out the deployment sequence determined and if the suitable launchers are available. This approach reverses the normal evolution of the launch and set-up procedure as it has been conceived up to now, which probably involves an intrinsic difficulty of implementation. On the other hand, to face the constellation launch and set-up problem using a different perspective could be conducive to identifying new needs as for the launcher capabilities and potential “gaps” to be filled in the launcher provider market.

Also the identification of some intermediate constellations can affect the set-up procedure: keeping in mind the progressive deployments determined, the analyst can try to target the sequence of launches aimed at filling the constellation planes to achieving an intermediate constellation at some deployment step.

To handle the intricate scenario of constellation set-up, it has been necessary to find an efficient formulation of the problem and to model the resolution methodology. To achieve this objective some assumptions and simplifications have been used. The fundamental assumption is that no multiple plane insertion capability is taken into account. This means that each launch is targeted to inject all the spacecraft lofted atop the launcher into the same orbital plane.

At the first-order level, the coverage performance can be considered as a general indicator of the service availability provided by the satellite system. The coverage performance in the selected region of primary interest is the figure of merit to assess the service capability of the system at each set-up step.

Study Case

The first step to cope with in the process of deploying a symmetric inclined constellation (like the Data Collection Constellation) is the exploration of potential intermediate deployment options. GMV has analyzed the partial deployments which allow achieving a certain service level in the primary area of interest, that is, the latitude band between 26° and 57° north. The mean revisit time in this latitude band is used as the figure of merit to test the quality of the intermediate constellations determined. Table 4 lists the main intermediate constellations identified while tracing the evolution of the system during the set-up phase.

Table 4: Intermediate Deployment Options for the Data Collection Constellation

Total number of s/c	**Number of planes**	**Phasing parameter**	**Mean revisit time (min)**
6	2	0	28.3
8	2	1	18.3
12	3	2	11.9
12	6	3	8.2
16	4	3	6.5
24	6	1	2.5

Each intermediate constellation is a Walker constellation identified by the total number of satellites (T), the number of the orbital planes and the phasing parameter (F). The relative spacing between satellites in adjacent planes is equal to F*(360 deg/T).

A criterion which can be used to select one or more partial deployments among the ones obtained is the mean revisit time that they provide. It is up to the analyst to decide if a degraded coverage performance is acceptable and, if so, which level of degradation is compatible with the mission requirements.

Populating the Data Collection Constellation with satellites involves carrying out a methodical sequence of launches into designated orbital planes. GMV has evaluated the optimal set-up strategy trying to trace the progressive deployments determined.

Three possible strategies are taken into account to deploy the constellation: the main steps of these set-up procedures coincide with the intermediate deployments outlined. The relevant information concerning each set-up strategy is presented in Table 5, Table 6 and Table 7, so as to allow comparing them in a straightforward way. The launches highlighted correspond to the achievement of one of the intermediate deployments identified. The number of satellites deployed after each set-up step is the sum of the satellites launched up to the last launch inclusive.

In these three tables a column indicates the coverage performance achieved at each set-up step, that is, after each launch. The evolution of the coverage performance during the deployment phase provides the coverage performance profile. The geographic coverage of the satellite system in the primary area of interest has been simulated over a period of 24 hours. The mean percentage of coverage over the whole simulation period is used as the fundamental figure of merit to test the service availability at each set-up step.

The coverage performance profile is basically the same for the set-up procedures analyzed. The deployment of the full 24-s/c constellation provides coverage of 94% of the selected latitude band on average. A total of respectively eight, six and seven launches were necessary to accomplish the constellation deployment by implementing these procedures. If the need to start the operational phase of the system and the revenue stream is a primary factor to determine the success of the mission, then the second strategy should be implemented. In fact, this is the fastest deployment, which involves the lowest number of launches.

Table 5: First set-up strategy for the Data Collection Constellation.

Launch number	Number of s/c launched per launch	Launcher used	Launcher cost ($M)	Coverage performance (%)	
1	3	Long March CZ-2C	15-20	15.3587	
2*	3	Long March CZ-2C	15-20	30.7116	*6 s/c in 2 planes*
3	4	Taurus	18-20	48.2016	
4	1	Pegasus XL	6-15	49.3200	
5*	1	Pegasus XL	6-15	56.0083	*12 s/c in 3 planes*
6*	4	Taurus	18-20	71.2554	*16 s/c in 4 planes*
7	4	Taurus	18-20	83.2483	
8*	4	Taurus	18-20	94.1819	*24 s/c in 6 planes*

* Denotes intermediate constellation as described in Table 4

Table 6: Second set-up strategy for the Data Collection Constellation

Launch number	Number of s/c launched per launch	Launcher used	Launcher cost ($M)	Coverage performance (%)	
1	4	Taurus	18-20	20.4945	
2*	4	Taurus	18-20	39.9056	*8 s/c in 2 planes*
3*	4	Taurus	18-20	56.1753	*12 s/c in 3 planes*
4*	4	Taurus	18-20	71.2554	*16 s/c in 4 planes*
5	4	Taurus	18-20	83.2483	
6*	4	Taurus	18-20	94.1819	*24 s/c in 4 planes*

* Denotes intermediate constellation as described in Table 4

Table 7: Third set-up strategy for the Data Collection Constellation

Launch number	Number of s/c launched per launch	Launcher used	Launcher cost ($M)	Coverage performance (%)	
1	4	Taurus	18-20	20.4945	
2*	2	Athena 1	16	30.2126	6 s/c in 2 planes
3	4	Taurus	18-20	48.0366	
4*	2	Athena 1	16	56.1753	12 s/c in 3 planes
5*	4	Taurus	18-20	71.2554	16 s/c in 4 planes
6	4	Taurus	18-20	83.2483	
7*	4	Taurus	18-20	94.1918	24 s/c in 6 planes

* Denotes intermediate constellation as described in Table 4

The average cost of the set-up strategies taken into account is estimated in Figure 3. It emerges that the second strategy, besides being the fastest, also involves the minimum total deployment cost. Therefore, it seems quite reasonable to select this strategy to set up the Data Collection Constellation.

It should be noted that the values of launch cost are approximations based on US Department of Transportation estimates or provided by the launcher manufacturers. These values are estimates only and may vary significantly depending on the mission.

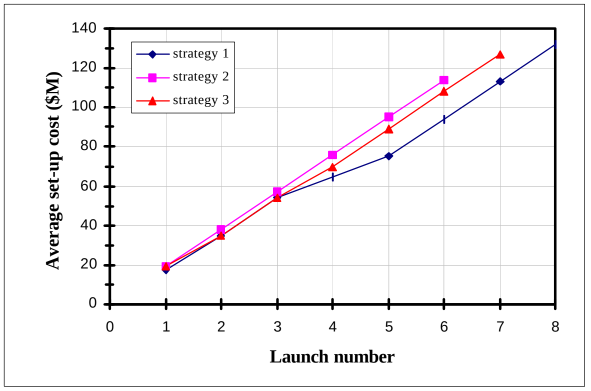

Figure 3: Average Set-up Cost for the Data Collection Constellation.

Conclusions for the Launch and Set-up Strategy

The analysis of the constellation launch and deployment strategies has been divided into the following investigation phases:

- Evaluation of the launch strategy (launch site selection and launcher selection)
- Evaluation of the injection and the transfer strategies
- Constellation set-up strategy.

It is unquestionable that the launch and deployment phase is a crucial step in achieving the full operational status of a constellation and plays an important role in the investment plan. The operational satellites must be deployed on schedule to catch the market, while trying to keep the launch cost, a significant portion of the overall system cost, to its lowest value.

On the basis of these considerations, the problem of deploying a satellite constellation has been tackled using a multi-criteria approach. The launch cost, the deployment duration and the service availability of the system emerged as the main factors which combine to determine the effectiveness of a set-up strategy.

The use of heavy lift vehicles to launch satellite constellations offers the potential advantage of reducing the deployment duration and the launch cost per satellite, thanks to the injection of multiple spacecraft per launch. On the other hand, in case of constellations of small LEO satellites, in some circumstances the big launchers may not be the most advantageous choice. Their mass injection capability, particularly targeted to heavy payloads and/or high orbits, cannot be used completely and in an effective way. These launchers are capable of launching an enormous number of small satellites into LEO orbits (sometimes more satellites that those of the whole constellation), which may not be a very interesting option unless a substantial multiple plane injection capability is made feasible. Small launchers can be more attractive to deploy LEO constellations of small satellites. The main drawback is that small launchers generally entail a rather high cost per s/c launched.

These observations seem to outline a “gap” in the launch capabilities provided by the state-of-the-art launch vehicle industry. The reduction of the launch cost and the development of a flexible injection capability could be a solution to fill this gap. The surge in space transportation demand is actually urging the launcher providers to enhance the capabilities of the existing vehicles and to develop new launchers to meet the emerging needs.

Replacement and Spare Strategy

Spare Policy Strategies.

Intrinsic Robustness to Satellite Failures

For some constellations, such as those providing telecommunications (e.g. Iridium, Teledesic, Globalstar) or navigation (e.g. GPS, GLONASS, GNSS) services, in order to avoid any deterioration of the service provided to the users, the constellation is designed to be "overpopulated". This means that the system is endowed with "spare" satellites already orbiting within the constellation.

Two main overpopulated configurations have been already implemented or are envisaged by the designers of the constellations that are in a development status:

– Overpopulated by One Satellite. The constellation designers purposely add one extra operational satellite per plane to the constellation, so that, if a satellite failure occurs, there is no time delay to replace the failed satellite. The constellation continues to function without interruption at maximum capacity with one satellite failure in an orbital plane. Telecommunications constellations are sometimes overpopulated by one satellite in order to provide redundancy in the system (i.e. Teledesic).

– Overpopulation by Two Satellites. In cases where the reliability and availability of the constellation are crucial, the constellation designers may overpopulate the constellation by two satellites per plane. This assures a service availability and reliability in over 99.99% of all situations. The reliability and availability of the constellation are only interrupted if there are three or more failures in the same plane at one time. This option is usually selected by navigation constellations (i.e. GPS and GLONASS) for whom reliability and availability of the signal are paramount.

In-orbit Spares

Other constellations allow that during a reduced period the constellation service may be "degraded" to a lower performance level. One can thus envisage that the constellation is functioning with a reduced number of in-orbit "spare" satellites.

The spare satellites are not exactly placed at the operational altitude of the constellation in order to avoid possible collisions of the non-controlled spares with the operational spacecraft. The slight difference in altitude implies a different inclination between the orbital planes of the spare and the operational satellites to counter-act the differential perturbations due to the Earth gravitational field and keep the nodes of the orbits to the same value.

To put a spare satellite into the right orbital slot, two Hohmann transfers can be performed, so as to re-acquire the nominal performance level. A first one to reach a phasing orbit and a second one to come back to the operational altitude. The phasing orbit is generally a circular orbit at a different altitude with respect to the constellation, so as to eliminate the phase difference between the spare and the operational satellites.

The time to replace a failed satellite by this method is usually on the order of few days (generally one or two). The cost drivers are at least the number of spare satellites, the repositioning ΔV (total and per satellite), the required time and the performance level.

Telecommunications constellations frequently use this strategy to replace failed satellites (i.e. Globalstar).

Spare Satellites in Parking Orbits

Another option consists in assuming that the "spare" satellites are orbiting in "parking" orbits. The constraints on such orbits are among others:

– the required ΔV to transfer from the parking orbit to any orbital plane of the constellation,

– the ΔV required for the satellite repositioning inside the constellation,

– the time delay between a satellite failure and the constellation orbital recovery.

The time to replace a failed satellite by this method is usually on the order of one to two months.

Telecommunications constellations usually use either this strategy or in-orbit spares to replace failed spacecraft (i.e. Iridium).

Spare Satellites on the Ground

The last option consists in keeping one or more "spare" satellites of the constellation on the ground. If we select a launch-on-demand as the replacement strategy, there are two possible options. Either a spacecraft is assumed to be stored on the ground and then launched as soon as the operator is able to secure a launcher for it. Or, the operator is able to manufacture a spacecraft quickly enough and secure a launch for it in order to replace the failed satellite.

Earth observation constellations sometimes use this strategy. Other types of constellations will sometimes use this strategy as a back-up to their primary replacement strategy (i.e. Globalstar which has on-orbit spares and ground spares which can be launched on demand).

This replacement strategy involves launching a spare satellite to replace a failed satellite only when the failure has occurred. Two conditions are mandatory to put forward this policy as feasible for a given mission: there must be a launcher with the capacity of carrying a single satellite in terms of mass and volume, and it is necessary an agreement between a launcher provider and the organism operating the constellation to ensure launch priority and availability. Because of these considerations, the time to replace a failed satellite by a launch on-demand is usually on the order of months up to one year.

Driving Parameters

The driving factors that determine the selection of a particular spare strategy are the service availability provided by the system and the satellite reliability over the constellation lifetime.

The ***reliability*** figure of a complex product like a satellite can be modeled using a function, R(t), which expresses the probability that the spacecraft is functioning in nominal conditions during a given time interval. R(t) can be computed using this equation:

$$R(t) = e^{(-t/MTBF)} \qquad (1)$$

R(t) is the reliability at time t and MTBF is the Mean Time Between Failures for the satellite. Given the value of the end-of-life reliability and the lifetime of the constellation, the MTBF can be computed and then the above equation can be used to evaluate the satellite reliability at each instant of time.

Taking into account the fact that no satellite has 100% reliability, the redundancy of the system should be optimized, either at the satellite level (i.e. satellite reliability, which involves higher manufacturing costs) or at the constellation level (i.e. multiple coverage). A trade-off between the satellite reliability and the level of constellation redundancy is to be accomplished versus the service availability required.

The ***performance level*** of the satellite system can be defined as the capability to meet the mission requirements, according to the mission profile. Performance parameters can be the geographic coverage, the mean or the maximum revisit time, etc. Based on this definition, the performance level achieved within a certain Mean Time To Replace (MTTR) proves to be a good indicator of the system ***service availability*** over the system lifetime. The nominal configuration of the constellation is designed specifically to fulfill the mission objectives and constraints. This means that a system that operates in the nominal status provides 100% performance level. When one or more satellite failures occur, we can expect that during a reduced period of time the constellation service may be "degraded" to a lower performance level, unless the constellation is overpopulated.

In order to estimate the evolution of the performance level over the system lifetime, first of all it is necessary to evaluate the instantaneous degradation of the performance level due to one or more satellite failures. To achieve this objective, we can simulate the failure of one, two and three satellites at the same time and calculate the performance level of the degraded configuration of the constellation. Note that the occurrence of three failures at the same time is a pessimistic condition, unless the number of satellites in the constellation is very high. The simulations of satellite failures must be enough to explore all the possible situations (for example, failures in the same orbital plane or in different orbital planes) and to allow averaging the values of the performance levels obtained. At the end of these simulations, we have computed the average performance levels with one, two and three simultaneous satellite failures.

Given these instantaneous performance levels, a Monte Carlo simulation is applied to calculate the mean performance level over the system lifetime for each of the possible replacement strategies outlined. It also calculates the average number of satellite failures during the constellation lifetime over all of the simulations. For those strategies which involve the use of a launcher, (i.e. launch-on-demand and perhaps the replacement for the overpopulated constellations), the average number of launcher failures can be estimated over all of the simulations. The launcher failure rate depends on the launcher reliability.

If a very high level of operational availability is required (e.g. 99.90% or more), spare satellites already located in orbit will be preferred. On the contrary, low level of requirements (e.g. less than 95%) will be compatible with a launch-on-demand strategy. In between, spare satellites in parking orbits will be preferred, and the number of spare orbital planes will depend on the availability requirement and on the altitude offset between the nominal and the parking orbits.

In general, the system operational availability is a key requirement for navigation and telecommunications constellations, thus leading to spare strategies based on in-orbit spares or spare satellites in parking orbits. On the other hand, no spacecraft back-up policy is generally envisioned for Earth observation constellations: when a satellite is no longer operative, there is no immediate replacement available. At best, a new spacecraft may be launched to replace the dead spacecraft within approximately one year.

Study Case

In the case of the Data Collection Mission, the requirements for service availability and reliability are compatible with a degraded service during a reduced period of time following one or more satellite failures. The acceptable duration of this period of lower performance depends strongly on the mission objectives and is a key factor to define the optimal replacement strategies. The MTTR a failed satellite is the parameter used to express the time constraints.

The mean performance level of the system has been evaluated over a lifetime of five years, which is the nominal lifetime of the Data Collection Constellation. Based on the review of analogous missions and satellite constellations, we have considered a satellite reliability of 0.8 and 0.6 at the end of the system lifetime. The satellite reliability at the end of the system lifetime has been used to parameterize the performance level versus the MTTR.

Three replacement strategies have been taken into account to accomplish this performance analysis:

- Launch on-demand
- Spare satellites in parking orbits
- In-orbit spare satellites.

The selection of a specific replacement strategy determines the time required to replace a failed satellite, and hence the achievable performance level of the system.

Figure 4 presents the mean performance profile versus the MTTR provided by the 24 satellites that comprise the constellation. This figure displays the system performance achieved implementing a launch on-demand or placing the spares in parking orbits.

The higher the satellite reliability, the higher the performance level of the system, because the failure probability and the failure rate are lower. This holds true independent of the specific replacement method selected. In addition, it is evident that the effect of the satellite reliability on the mean performance of the system increases as the MTTR increases. This is due to the higher failure probability associated with lower satellite reliability.

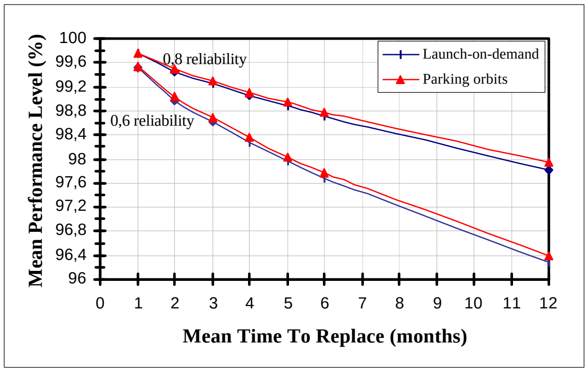

Figure 4: Mean Performance Level of the Data Collection System versus MTTR

The MTTR being equal, the performance levels provided by the launch-on-demand strategy and the spares in parking orbits are practically the same. The selection of the launch-on-demand strategy entails a slightly worse performance because of the impact of the launcher reliability: the performance level is not only affected by the satellite failure rate, but also by the launch failure rate. If the launch of a spare satellite fails, a new launch must be arranged and performed, with the consequent delay in the replacement of the failed spacecraft. This negatively affects the performance level of the system versus time. A mean launcher reliability of 0.9 has been assumed. This value is to be considered as a reasonable estimation, since the determination of reliability can be difficult or impossible for relatively new or untested launch systems.

Besides the two replacement strategies presented in Figure 4, in-orbit spare satellites have also been taken into account. In this case, the MTTR is on the order of few days (generally one or two). The corresponding mean performance level of the system is:

– about 99.92% for a satellite reliability of 0.8,

– about 99.85%, for a satellite reliability of 0.6.

If a very high level of operational availability is required, spare satellites already located in orbit will be preferred, since this strategy allows maintaining a high performance level thanks to a reduced MTTR. The in-orbit spare strategy allows replacing quickly a failed satellite for reasonable fuel consumption by deploying at least one spare satellite per orbital plane. The advantages of an in-orbit spare strategy should be mitigated by the fact that it requires at least one additional satellite per orbital plane, which may have a strong impact on the launch and deployment of the constellation. In the case of the 24-s/c constellation, a minimum of six spare satellites (1/4 of the number of the operational satellites) should be placed in orbit. Unless the requirements for availability and reliability of service are very demanding, other replacement strategies should be investigated, as we cannot rule out that they prove to be a better trade-off between the number of spare satellites required and the degradation of the service.

Lower performance levels and longer MTTR values are compatible with a launch-on-demand strategy and/or with spare satellites in intermediate parking orbits. One point should be highlighted as regards these replacement strategies. The set-up and launch to replace a failed satellite imply a MTTR that is generally longer than in the case of spare satellites in parking orbits. This means that, even if the performance profiles provided by these two strategies are very similar, the MTTR is likely to be different, which leads to different effective performance levels. Actually, the agreement between the constellation operators and the launcher providers dictates the real conditions of launch availability and priority. Therefore, the MTTR associated with a launch on-demand may vary considerably according to the real scenario taken into account.

From the evaluation of the replacement strategies outlined, it emerges that the performance level provided by spare s/c in parking orbits is rather good if the MTTR can be kept to a low value (one/two months). Thus, it is interesting to investigate this strategy in more depth in order to assess its feasibility and the potential advantages that it can offer.

Figure 5 shows the required circular drift orbit altitude versus the MTTR for different numbers of spare planes (assuming that the drift orbit is circular).

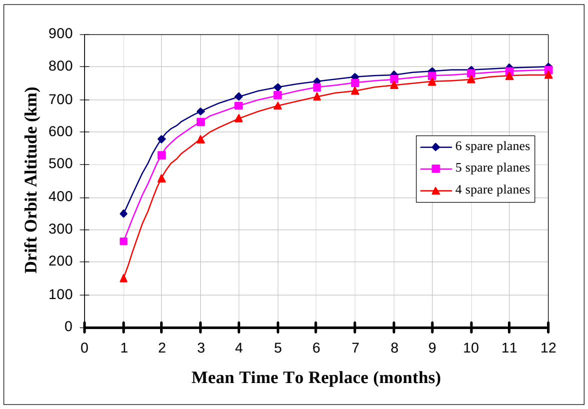

Figure 5: Drift Orbit Altitude versus MTTR for the Data Collection Mission

The mean time to replace a failed satellite is the sum of the time needed for the nodes to coincide, and the duration of the altitude-raising phase. The latter phase brings a significant contribution to the overall MTTR in case of electric propulsion, while almost negligible in case of chemical propulsion. If we assume to use a chemical propulsion system to perform the impulsive orbit maneuvers, the MTTR is a direct function of:

- the number of spare orbital planes, which determines the maximum initial nodal angular difference between the orbital planes of the spare and the failed satellites;

- the altitude of the spare orbit, which determines the relative nodal drift rate between the orbital planes of the spare and the failed satellites.

The initial angle between the orbital planes of the spare and the failed satellites is uniformly distributed between zero and the angular difference between two consecutive spare planes (worst case corresponding to the maximum drift duration). Thus, the time for the nodes to coincide (the MTTR) is half this maximum drift duration (the time spent on the spare orbit).

The selection of the number of spare planes is mainly driven by two factors: the desired MTTR for a given performance level and the number of spare satellites to be injected into orbit. The more spare planes, the lower the MTTR and the higher the number of additional satellites to be deployed.

The cost of the propulsive maneuvers to perform the transfer from the spare orbit to the operational orbit of the constellation can be evaluated using Figure 1 (based on the assumption of using chemical propulsion).

To complete the analysis of the replacement strategies, GMV has estimated the average number of satellite failures during the constellation lifetime. This parameter depends directly on the satellite reliability and on the system lifetime. The results obtained are presented in Table 8. The satellite failures during the constellation lifetime are expressed as percentages of the number of satellites of the constellation.

Table 8: Statistics of Satellite Failures

System lifetime (years)	Satellite reliability	Average percentage of failures during the constellation lifetime
5	0.6	39.2%
5	0.8	19.7%

These results are to be considered as global statistical estimates, whose objective is to provide a comprehensive indication of the system robustness.

Conclusions for the Replacement Strategy.

Of the three replacement strategies studied (in-orbit spares, spares in a parking orbit, and launch-on-demand), the best strategy from the point of view of maintaining the constellation performance is always going to be in-orbit spares. However, depending upon the performance requirements of the constellation, in-orbit spares may provide a higher than needed constellation performance at a much higher monetary cost. Concretely, if there are many orbital planes, this presumes adding at least one additional on-orbit spare satellite to each plane, which may entail many extra satellites to be built and launched.

In some cases, the replacement strategies of spares in parking orbits and launch-on-demand may become very interesting. In particular, having spares in a parking orbit is a viable replacement strategy when the parking orbit is not very low in altitude, and hence will not be overly affected by atmospheric drag. Additionally, if the number of spare planes is high, the time to replace a failed satellite is lower, and therefore this strategy becomes more attractive. However, with more spare planes, more satellites need to be built and launched.

The launch-on-demand replacement strategy is particularly useful when the constellation performance requirements are not strict. This option generally takes the most time in order to replace a failed satellite. However, in the long run, it is usually the cheapest. Additionally, it should be noted that the amount of time it takes to replace a failed satellite with this strategy depends upon the availability of appropriate launchers. If no appropriate launchers are immediately available, this strategy can take a long time to replace a failed satellite. This strategy often works very well as a back-up in combination with either one of the two previous replacement strategies.

Keeping in mind the advantages and the drawbacks of each replacement strategy, the constellation operators select the spare policy so as to comply with the requirements in terms of system availability.

Constellation End-of-Life Policy

The concept of End-Of-Life (EOL) for a constellation is somewhat different than the concept of end-of-life for a single spacecraft. Typically, a single spacecraft is used until the absolute very end of its functional lifetime. This is typically until the absolute end of its on-board fuel required for orbit maintenance, or until the power source on the spacecraft fails, or all the on-board instruments fail. This is not necessarily or even generally true for spacecraft in constellations. Since constellations usually depend on having all satellites completely functional, if a spacecraft in a constellation fails even partially and cannot be completely revived, it may be replaced by a new spacecraft, even if some functionality still remains.

In January 1998, the US Department of Defense (DoD) came out with a disposal policy for DoD spacecraft in all types of orbits[9]. In the future, US DoD LEO spacecraft will all be positioned so as to burn up on re-entry into the atmosphere within 25 years after the end of the operational lifetime. US DoD GEO spacecraft will be maneuvered into a graveyard orbit approximately 300 km above geostationary altitude at 36100 km. Of particular interest was the first articulated graveyard orbit for MEO spacecraft. The US DoD defined two broad graveyard orbits for the disposal of MEO spacecraft. MEO spacecraft may either be placed in circular graveyard orbits of between 2000 km and 19700 km in altitude, or between 20700 and 35300 km in altitude. In addition, the US DoD has articulated a policy of a graveyard orbit for its own GPS satellites (semi-synchronous orbits at about 20000 km of altitude). This graveyard orbit is between approximately 500 km above semi-synchronous and 500 km below synchronous. It should be stressed however, that this policy currently applies to only US DoD satellites and is not an international standard.

The orbit altitude of the constellation (or the perigee and apogee altitudes in case of constellation of satellites in elliptical orbits) strongly affects the identification of an effective EOL strategy. Also the cost of the EOL strategy, in terms of the allocated fuel budget, is mainly determined by the altitude band in which the constellation is arranged.

End-of-Life Strategies for LEO Constellations

The orbital lifetimes of satellites passing through the densely populated LEO band can be several thousands of years for near circular orbits of high mean altitude (up to 2000 km). Thus, it is imperative to define a limitation of the orbital lifetime for spacecraft arranged in LEO with large masses and cross sections, so as to avoid the accumulation of large objects and the further growth of small debris due to explosions and collisions. An upper limit of 25 years for remaining orbit lifetimes after mission completion is currently being discussed at international level as a design guideline for operators of satellites or constellations that cross the LEO band. Such EOL policy can be implemented in future designs by means of active de-orbiting into a direct or delayed re-entry trajectory, or by taking advantage of natural perturbations via properly selected EOL initial orbit conditions.

As part of the active or natural orbit lifetime reduction, various aspects need to be considered in depth. First of all, the technical and economical feasibility of active de-orbiting in terms of propellant mass penalties and required attitude and orbit control sub-system enhancements. Second of all, the resulting residual risk on the ground due to spacecraft fragments which could survive the re-entry phase.

The parameters that can be controlled to affect the orbital lifetime of a satellite at the end of its operational phase are the following:

– the ΔV required to perform a de-orbiting maneuver into a direct re-entry trajectory or into a self-decaying orbit with a reduced lifetime
– the deployment of drag augmentation devices or the acquisition of a high drag attitude.

Controlled De-Orbiting Maneuver

Immediate de-orbiting at mission completion entails carrying out a maneuver that leads to steep, controlled re-entry and burn-up in the atmosphere, or to ground impact at a pre-assigned, safe location. The initiating de-orbiting maneuver must be implemented such that the resulting perigee altitude is sufficiently low to enable the atmospheric capture of the spacecraft ideally within one revolution. In any case, in order to maintain control over the re-entry location, the time between the de-orbiting maneuver and the final re-entry should not exceed one to two orbits.

In order to initiate a controlled re-entry, a sufficient large ΔV maneuver at apogee must be applied to lower the perigee altitude well into the atmosphere. The selection of the de-orbiting perigee altitude depends on the mass-to-area ratio and on the drag coefficient of the re-entering spacecraft. For non-lifting bodies, a perigee altitude of about 60 km ascertains a safe capture of the spacecraft in the lower atmosphere, followed by an immediate, steep re-entry with a well-controlled and relatively small impact for the fragments that survive the deceleration and heating. Heavy, compact spacecraft may require lower perigee altitudes.

Figure 6 shows the ΔV required to perform a controlled de-orbiting of a spacecraft from a near-circular orbit of given initial altitude into a direct re-entry orbit with a perigee height of 60 km. In Figure 7 the equivalent propellant mass fraction is provided for exhaust velocities of 2000 m/s (mono-propellant hydrazine system of specific impulse $I_{sp} \cong 220$ s) and 3000 m/s (bi-propellant hydrazine / N_2O_4 system of specific impulse $I_{sp} \cong 310$ s). The propellant mass fraction m_p / m_0 is related to the necessary ΔV by the rocket equation:

$$m_p / m_0 = 1 - e^{-\Delta V/Ve} \qquad (2)$$

where:

m_p / m_0 = fuel mass fraction of the EOL s/c mass

ΔV = velocity impulse required for controlled direct de-orbiting (m/s)

Ve = exhaust velocity of the spacecraft thrusters at EOL (m/s).

In the case of the Data Collection Constellation, an immediate de-orbiting maneuver from the constellation nominal altitude (850 km) would entail a ΔV of about 215 m/s and a mass fraction between 0.07 ($I_{sp} \cong 220$ s) and 0.1 ($I_{sp} \cong 310$ s).

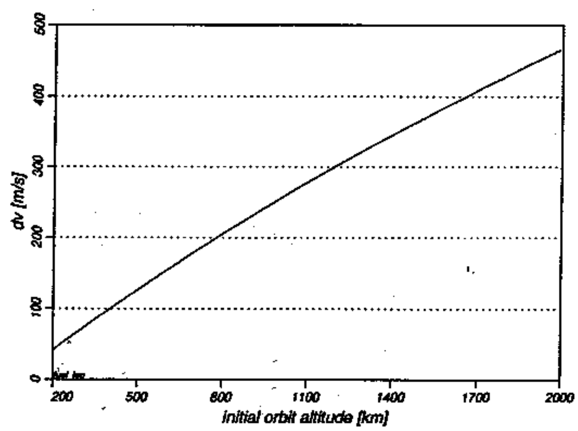

Figure 6:DV for a controlled de-orbiting of LEO s/c

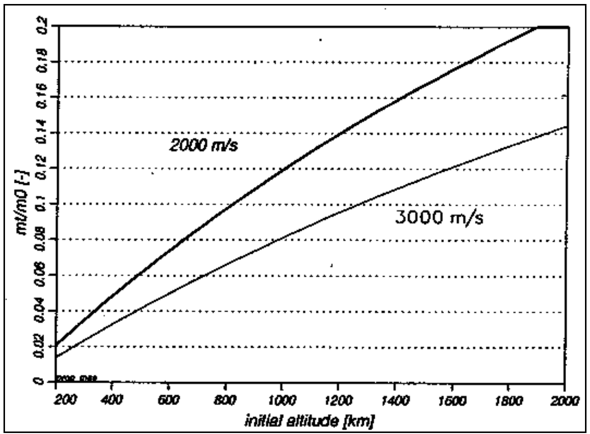

Figure 7: Propellant mass fraction for controlled de-orbiting of LEO spacecraft from near-circular orbits.

The very high ΔV and pointing requirements for an immediate de-orbiting maneuver often exceed the capabilities of a spacecraft after mission completion, because they may be too expensive and technically demanding to be implemented in the system design.

Uncontrolled De-Orbiting Maneuver

An uncontrolled de-orbiting of a satellite at the end of its operational lifetime is initiated by one or several short arc maneuvers at apogee passes (for mono- or bi-propellant systems), or by an extended low-thrust maneuver (in case of ionic thrusters). The objective is to reduce the perigee altitude of the orbit and to cause a decay and final re-entry within a limited time span. In order to reduce the number of high mass LEO objects with considerable cross sections, the spacecraft and constellation operators seem to converge to a commonly accepted lifetime limitation of 25 years for the EOL disposal orbits of their LEO missions[8, 9]. This EOL strategy is intended to narrow down the risk of future explosions and collisions in the LEO region, a risk that cannot be neglected because of the catastrophic consequences it may have.

Following the de-orbiting of a spacecraft into a reduced lifetime orbit, the altitude decay and the final re-entry footprint will be determined by highly variable aerodynamic drag forces. The uncertainties in the orbit and attitude prediction are dominated by the uncertainties in the atmospheric model, and in the solar and geomagnetic activity forecasts. In consequence, the re-entry time and location cannot be predicted reliably until a few revolutions before the final descent. An uncontrolled re-entry can occur anywhere in a latitude band $\phi \leq \pm i$ (where i is the orbit inclination). To minimize the risk due to ground impact of surviving fragments, the mission operators should assess the consequences of the break-up and incomplete burn-up of the spacecraft caused by atmospheric friction. Typical break-up altitudes for uncontrolled re-entries are around 80 km.

Figure 8 displays the propellant mass fraction required to de-orbit a spacecraft from a given near-circular orbit into a reduced lifetime orbit which will re-enter within 25 years. This figure highlights the dependence of the propellant mass fraction on the efficiency of the propulsion system (expressed in terms of the thruster exhaust velocity). Here 16000 m/s and 30000 m/s correspond to two different ion thruster systems.

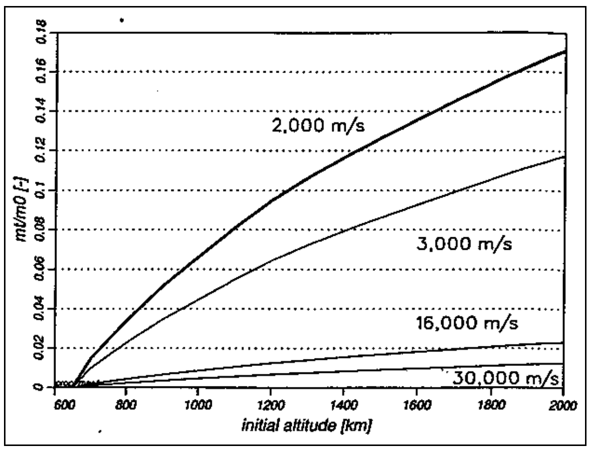

Figure 8 Propellant mass fraction for an uncontrolled de-orbiting from a near-circular LEO orbit into an eccentric orbit of reduced lifetime (< 25 years).

For the Data Collection Mission, the mass fraction in case of uncontrolled de-orbiting would be on the order of 0.03-0.04, assuming a mono- or bi-propellant propulsive system.

The uncontrolled de-orbiting EOL strategy entails a ΔV requirement that is considerably lower that the value determined for a controlled de-orbiting. Apart from this advantage, it is to be noted that an uncontrolled de-orbiting into reduced lifetime orbits should be avoided in the case of hazardous payloads.

Non-Propulsive Orbit Lifetime Reduction

The following non-propulsive procedures could be applied to reduce the orbital lifetime of large spacecraft after their mission completion (EOL).

- EOL orbit altitude. Leaving a satellite in a near-circular orbit of low mean altitude (e.g. H_{mean} < 500 km), or in a highly eccentric orbit of very low perigee altitude (e.g. H_{pe} < 200 km), will in most cases result in a residual lifetime of less than 25 years. The remaining time in orbit is driven by air drag, which is directly proportional to the area-to-mass ratio of the spacecraft, and to the local air density (which mainly depends on the solar activity in the course of the 11-year solar cycle). For highly eccentric orbits, luni-solar perturbations are key factors.

- EOL drag augmentation. Air drag and solar radiation pressure forces are directly proportional to the spacecraft's area-to-mass ratio. The effective cross section at EOL could be increased by the inflation of a balloon-type structure, or by the deployment of an umbrella-type light-weight appendage with a large cross section. The increased air drag directly reduces the orbital lifetime. On highly eccentric orbits, solar radiation pressure could furthermore assist in lowering the perigee altitude, which entails a corresponding increase of the drag perturbation.

- EOL orbit orientation. An optimized selection of the orbital plane orientation at mission completion proves to have a strong influence on the residual lifetime of a satellite, particularly for HEO and GTO orbits. Depending on the season of the year, and the node position (RAAN), the HEO or GTO perigee decreases or increases in consequence of a periodic perturbation, whose amplitude depends on the inclination of the orbital plane and on the eccentricity of the orbit. The amplitude of this perigee variation can range over more than 100 km, and it can lead to an atmospheric capture or even to an intersection with the Earth ellipsoid, thus causing re-entry.

All non-propulsive lifetime reduction measures aim at an accelerated conversion of orbital energy into heat (air drag). The resulting orbit decay is gradual, leading to a natural re-entry with little control of the impact footprint.

End-of-Life Strategies for GEO and MEO Constellations

Not all space missions allow carrying out an immediate or delayed de-orbiting of spacecraft at the end of their operational lifetime. Particularly satellites located near the GEO orbit, or satellites in semi-synchronous orbits (used by the GPS and the GLONASS navigation and positioning constellations) cannot be de-orbited. This is due to the unrealistic requirements in terms of ΔV and fuel budget which would be necessary for a direct, as well as for a delayed de-orbiting maneuver after mission completion.

In case of GEO satellites, a ***disposal orbit*** (also called "***graveyard orbit***") above the GEO ring has already been used by many spacecraft operators, so as to reduce the collision risk at the operational GEO altitude. The altitude-raising phase should be to a disposal orbit whose perigee altitude is at least ΔH_{GEO} km above the GEO ring. This minimum clearance altitude increase has been agreed by the IADC[8] (Inter-Agency Debris Committee) Members, and it is defined as:

$$\Delta H_{GEO} = 235 + 1000*(C_r)_{max} * (A / m)_{max} \qquad (3)$$

where:

ΔH_{GEO} = minimum perigee altitude clearance above GEO (km)

$(C_r)_{max}$= maximum solar radiation pressure coefficient

$(A/m)_{max}$ = maximum area-to-mass ratio of the satellite (m^2/kg).

C_r can take values between 0 and 2. A realistic value for GEO satellites would be $1 \leq C_r \leq 2$.

This recommended clearance is based on the analysis of the long-term orbit evolution of a graveyard orbit under the influence of gravitational forces (mainly due to the Sun, the Moon and the oblate Earth), and of solar radiation pressure forces. In order to make the most efficient use of available fuel resources, the re-orbiting to a disposal orbit should be performed via multiple burns, based on the best fuel estimation, so that the final orbit is near-circular and the propellant mass is depleted. Subsequently, the spacecraft should be switched off in a controlled manner, with dissipation of all on-board energy sources (fuel, pressure devices and batteries).

As a rule-of-thumb, a total velocity impulse of ΔV = 3.64 m/s is required for each circular orbit increase by ΔH = 100 km (i.e., $\Delta V \cong 11$ m/s to achieve the recommended altitude increase of $\Delta H > 300$ km above the GEO altitude).

The use of disposal orbits for other altitude bands has not been recommended yet. The definition of an internationally accepted policy for possible graveyard orbits for MEO satellites is still an open issue.

Conclusions for the End-of-Life Policy

The issue of EOL strategies for constellations has come to the fore following the recent development and the expected surge of multi-satellite systems. It is no longer possible to neglect the problems and the risks stemming from the considerable number of orbiting spacecraft positioned in the most attractive altitude bands. Consequently, an increasing interest in the EOL policy has arisen at international level, fostering the analysis of possible solutions.

The review of the possible constellation EOL policies has pointed out that the driving factor in the selection of a particular strategy is the satellite altitude. First of all, the satellite altitude has a strong impact on the determination of the relevant orbital perturbations that affect the system. Eventually, these perturbations could be used to de-orbit a LEO constellation, thus assisting the EOL strategy implemented. Second of all, the cost of the EOL maneuvers depends on the EOL altitude of the satellites. This holds true both in case of de-orbiting maneuvers to force the re-entry of LEO spacecraft and in case of maneuvering GEO satellites to graveyard orbits above the GEO altitude.

The effectiveness of an EOL strategy is the result of a trade-off between the cost of the required maneuvers and the collision risk mitigation obtained via the selected EOL procedure. For LEO satellites, a fundamental constraint is the residual lifetime after the completion of the operational phase of the system. An upper limit of 25 years for post-operation lifetimes is currently being put forward as a design guideline for operators of satellites or constellations that cross the LEO band.

The analysis of the possible end-of-life strategies has highlighted the need to define an internationally accepted regulation as far as the EOL policy is concerned. In addition, a big gap needs to be filled: the definition of feasible and effective EOL strategies for satellites arranged in the MEO altitude band.

Conclusions

The launch and set-up, the replacement of failed satellites and the end-of-life policy are key steps in constellation development. In order to outline a general and effective approach to handle these phases, GMV has accomplished a study of the fundamental parameters affecting each step and has assessed their impact on the system management plan.

The general concepts and trade-off results have been tested taking into account a constellation of small satellites and using in-house software tools and algorithms. This has allowed pointing out some interesting trends and controversial issues that characterize the emerging satellite constellations.

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